Launching Small Satellites on the H-IIA Rocket

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Abstract. Over the last 40 years, the lowest cost transportation to space for small payloads has been achieved by utilizing excess capacity on large launches. Because the mass and volume used by small payloads on large vehicles are otherwise unused, often the pricing only needs to cover manifesting, integration and qualification of the so-called piggyback payload. Unfortunately this lack of economic incentive has suppressed supply – large rocket developers and operators may have little or no incentive to invest in small payload accommodations.

In taking on the management of Japan's newest and largest rocket, the H-IIA, from the National Space Development Agency of Japan (NASDA), Mitsubishi Heavy Industries has decided to aggressively pursue opportunities for comanifesting nano, micro and mini-satellites, leveraging proven capabilities developed by NASDA for small satellite launches on Japan's largest launch vehicle. This new capability, to be demonstrated on the next H-IIA launch this year, has the potential to more than double the secondary space available to small payloads worldwide, and will reduce queueing time for launching small satellites.

Introduction

The current state of the space industry is poor. There is little demand for telecom satellites, limited usage of the International Space Station, and minimal advanced vehicle development projects underway. In this situation, small satellites have the potential to help rejuvenate the space market. In general, small satellites do not require significant budgets and they can be developed with little technical difficulty, if a customer so chooses. These factors induce many newcomers to enter the small satellite world, with a variety of new, innovative ideas for small satellite applications.

Mitsubishi Heavy Industries (MHI) is aiding the small satellite community by preparing to offer piggyback launch accommodations and services to small satellite developers. MHI plans to provide launches to Sun-

Synchronous Orbits (SSO) and other Low-Earth Orbits (LEO), as well as Geosynchronous Transfer Orbit (GTO). Accommodations range from approximately 50 kg to over 300 kg. AeroAstro and AeroAstro Japan are cooperating with MHI to supply sales, marketing, and engineering support to their secondary payload program.

This paper outlines the launch services offered aboard the H-IIA for small, piggyback payloads. In particular, the accommodations for a sample 50 kg satellite are outlined in detail.

H-IIA Launch Vehicle Flight Results

After two successful test flights and two operational flights, the H-IIA launch vehicle lifted off on its third operational mission on March 28, 2003. The H-IIA launches have brought a total of eleven payloads to their desired orbits on time. Four of the eleven payloads were "small satellites". Table 1 shows the flight results of the H-IIA launch vehicle to date. The data demonstrates not only the reliability but also the accurate performance of the H-IIA launch vehicle.

Basic Concept of Piggyback Launch on the H-IIA

The accommodations and services offered by the H-IIA launch vehicle to a small, piggyback satellite depends heavily on that of the primary payload. Nevertheless, H-IIA aims to provide the small satellite developer with the same level of service as the primary payload, without disrupting the mission of the primary payload. A variety of services can be requested, and H-IIA will make every effort to address these requests. Detailed discussions regarding specific accommodations and services are held on a case-by-case basis prior to negotiating the launch service contract.

Table 1. Flight Results of the H-IIA Launch Vehicle.

<table>
<tr><th>Launcher</th><th colspan="2">TF1
(H2A202)</th><th colspan="2">TF2
(H2A2024)</th><th colspan="2">F3
(H2A2024)</th><th></th><th colspan="2">F4
(H2A202)</th><th>F5
(H2A2024)</th></tr>
<tr><td>Launch date</td><td colspan="2">August 29, 2001
</td><td colspan="2">February 4, 2002
</td><td colspan="2">September 10, 2002
</td><td rowspan="4"></td><td colspan="2">December 14, 2002
</td><td>March 28, 2003
</td></tr>
<tr><td>Satellite</td><td colspan="2">LRE</td><td colspan="2">MDS-1
DASH</td><td colspan="2">DRTS
USERS</td><td colspan="2">ADEOS-II
FedSat
WEOS
μ-LabSat</td><td>Government Mission (Dual)</td></tr>
<tr><td>Payload Fairing</td><td colspan="2">4S</td><td colspan="2">4/4D-LC</td><td colspan="2">4/4D-LC</td><td colspan="2">5S</td><td>4/4D-LC</td></tr>
<tr><td>Orbit</td><td colspan="2">GTO</td><td colspan="2">GTO (MDS-1)
LEO (DASH)</td><td colspan="2">GTO (DRTS)
LEO (USERS)</td><td colspan="2">SSO</td><td>SSO</td></tr>
<tr><td></td><td>Achieved (Target)</td><td>Δ</td><td>Achieved (Target)</td><td>Δ</td><td>Achieved (Target)</td><td>Δ</td><td></td><td>Achieved (Target)</td><td>Δ</td><td></td></tr>
<tr><td>Apogee Altitude (km)</td><td>36,191
(36,186)</td><td>5</td><td>35,696
(35,735)</td><td>-39</td><td>36,203
(36,206)</td><td>-3</td><td>Semi-Major Axis (km)</td><td>7,190
(7,190 ±20)</td><td>0</td><td rowspan="3">N/A</td></tr>
<tr><td>Perigee Altitude (km)</td><td>251
(251)</td><td>0</td><td>500
(500)</td><td>0</td><td>449.5
(450.0)</td><td>-0.5</td><td>Eccentricity</td><td>0.001
(0.001 -0.001 +0.003)</td><td>0</td></tr>
<tr><td>Orbital Inclination (deg)</td><td>28.50
(28.49)</td><td>0.01</td><td>28.5
(28.5)</td><td>0.0</td><td>28.5
(28.5)</td><td>0.0</td><td>Inclination</td><td>96.68
(98.67± 0.25)</td><td>0.01</td></tr>
<tr><td>Argument of Perigee (deg)</td><td>179.2
(179.1)</td><td>0.1</td><td>179
(179)</td><td>0</td><td>179
(179)</td><td>0</td><td colspan="4"></td></tr>
</table>

(Courtesy of NASDA)

Case of 50 kg Satellite

In order to demonstrate the capabilities of the H-IIA for launch of small satellites using piggyback launch accommodations, an example has been selected, studied, and detailed in the sections below. This example is for a 50 kg satellite with main dimensions of 500mm x 500mm x 450mm and a top protrusion with dimensions of Ø225mm x 50mm.

Services Available for Small Satellite

The following services are provided by H-IIA to the small satellite:

- ***Separation of Satellite:***

  In order to be separated, three split springs are to be installed at the separation portion of the satellite. The maximum stress limit at the edge of the split spring is 500N.

- ***Command Signal to Initialize Satellite Equipment:***

  A command to initialize equipment onboard the satellite can be made by the separation detector. Though the provision of the metal adapter on the rocket side is included as part of the launch services, the separation detector must be provided by the satellite developer.

- ***Umbilical***

  Umbilical (electrical) support is provided for use by the small satellite. The level of umbilical support available is dependent on the requirements of the primary payload.

- ***External Battery Charging***

  If the small satellite developer prepares a cable from the satellite to the exterior door of the rocket, the satellite battery can be charged from outside its fairing.

Launch Environment

Mechanical Environment

Quasi-Static Acceleration

Table 2 shows the maximum (three-sigma) quasi-static acceleration in both the longitudinal and lateral directions. The maximum quasi-static acceleration – a combination of static acceleration and low-frequency dynamic acceleration – is experienced at main engine (of the first stage) cutoff. The satellite structure must be designed to withstand this maximum aceeleration.

Table 2. Quasi-Static Acceleration.

	Longitudinal Acceleration	Lateral Acceleration
Compression	-58.84 m/s^2 (-6.0 g)	±49.04 m/s^2 (±5.0 g)
Tension	49.04 m/s^2 (5.0 g)	±49.04 m/s^2 (±5.0 g)

Sine Wave Vibration

Table 3 shows the maximum (three-sigma) sine wave vibration limit levels in both the longitudinal and lateral directions. These levels are prescribed at the satellite interface (satellite separation plane). The vibration is applied at the base of the adapter with a 4 octave/minute sweep rate in the up and down direction so that vibration levels at the spacecraft interface are equal to the above levels.

Table 3. Sine Wave Vibration Levels.

Direction	Frequency (Hz)	Acceleration
Longitudinal Direction	5-100	24.52 m/s^2 (2.5 g)
Lateral Direction	5-100	19.62 m/s^2

Random Vibration

Table 4 shows the random vibration levels in both the longitudinal and lateral directions. Random vibration is primarily caused by acoustic noise. The satellite must be able to endure this vibration environment for 60 seconds.

Table 4. Random Vibration Levels.

Frequency Width (Hz)	Acceleration (G^2/Hz)
20-200	+3 dB/octave
200-2,000	0.032
Actual	7.8 Grms

Acoustics

Table 5 shows the acoustic environment levels (two-sigma). Random vibrations are generated by the noise of the first stage main engine and Solid Rocket Booster, and the pressure vibration caused by buffeting and boundary layer noise during the phase of the transonic flight and the high dynamic pressure. The satellite must be able to endure this environment for 80 seconds.

Table 5. Acoustic Pressure Inside Fairing (No Acoustic Blanket)

Center Frequency (Hz)	H2A202 Acoustic Level (dB)	H2A212 Acoustic Level [Reference] (dB)
31.5	125.0	128.0
63	126.5	129.5
125	132.0	135.0
250	136.0	139.0
500	134.0	137.0
1,000	136.0	139.0
2,000	130.0	133.0
4,000	121.0	124.0
8,000	116.0	119.0
Overall	141.5	144.5

Shock

Pyrotechnic shocks are imparted to the satellite at several different points during the flight: separation of the fairing, separation of the first and second stages, and separation of the satellite from the Payload Attached Fitting (PAF). The largest shock the satellite experiences is due to the cutting of the clamp band at separation from the PAF.

Thermal Environment

Prelaunch Thermal Environment

The air inside the fairing is conditioned to varying degrees during the various stages prior to launch. The temperature inside the fairing is set by the temperature required by the primary payload, and can range from 10 to 25C. The maximum flow rate is 100 Sm^3/min. Note, however, that in the case of a dual launch, the maximum flow rate is 50 Sm^3/min per satellite. The relative humidity during the maintenance periods prior to launch is 40-50%. The relative humidity after the access work is complete but prior to launch is 0-50%. Throughout the prelaunch phase, a Class 5000 cleanliness is maintained.

Flight Thermal Environment

The satellite is protected by the payload fairing during initial ascent, but aerodynamic heating causes the satellite to receive a time-dependent radiation heating environment from the internal surface of this fairing prior to fairing jettison. The maximum heat flux radiated by the fairing is less than 500 W/m^2. The maximum temperature inside the fairing is 110C at the internal surface of the cylinder section of the fairing. The aerodynamic heat acceleration after separation of the fairing from the launch vehicle is less than 1,135 W/m^2.

The temperature at the connection between the PAF and the satellite varies between –10C and 50C.

Internal Pressure Environment

The air inside the payload fairing is vented during the ascent phase through one-way vent ports. The pressure decay rate typically varies while the launch vehicle is in transonic flight, with a maximum rate of 4.51KPa/s.

Interface Conditions

Mechanical Interface

The satellite must fit within a main envelope of 500mm x 500mm x 450mm plus a top protrusion of Ø225mm x 50mm. The center of gravity (CG) offset must be 250mm or less in longitudinal direction and 25mm or less in the orthogonal directions. The moment of intertia must be 1kgm^2 or less in each axis.

The fundamental frequency of the satellite must be less than 100 Hz in the longitudinal direction and less than 50 Hz in the lateral direction.

The satellite and the rocket are connected by the PAF with a clamp band. The tension in the radial direction at the connection with the clamp band is 6375 ±486 N (650 ±50 kgf).

Electrical Interface

There are 10 discrete command signals available from the rocket to the satellites – these must be shared between the primary and the piggyback satellite. The main electrical characteristics of these signals include:

- Voltage: +24V to +34V
- Current: less than 150 mA
- Supply time (duration): 500 ±50ms

No telemetry signal can be exchanged between the rocket and the satellite.

The number of electrical umbilicals available to the small satellite is limited by the number required by the primary payload.

If the satellite developer prepares a cable from the satellite to the operation door of the rocket, the satellite battery can be charged from outside its fairing.

Orbit Insertion Conditions

The orbit insertion of the piggyback satellite is dependent on that of the primary payload. As a reference, the typical orbit insertions of the primary payload and its accuracy (3-sigma) are shown below.

For a Geosynchronous Transfer Orbit (GTO), the typical orbital parameters are as follows:

- Apogee Altitude: 36.226 km
- Perigee Altitude: 250 km
- Angle of Inclination: 28.5 deg
- Argument of Perigee: 179.0 deg

For the same GTO case, the corresponding injection accuracies are as follows:

- Apogee Altitude Accuracy: ±180 km
- Perigee Altitude Accuracy: ±4 km
- Angle of Inclination Accuracy ± 0.02 deg
- Argument of Perigee Accuracy: ±0.4 deg
- Longitude of Ascending Node Accuracy: ±0.4 deg

For a Sun-Synchronous Orbit (SSO), the typical orbital parameters are as follows:

- Circular Orbit Altitude: 800 km
- Angle of Inclination: 98.9 deg
- Eccentricity: 0 – 0.001

For the same SSO case, the corresponding injection accuracies are as follows:

- Semi-Major Axis Accuracy: ±10 km
- Angle of Inclination Accuracy: ±0.18 deg
- Longitude of Ascending Node Accuracy: ±0.1 deg

Required Environmental Tests

The satellite developer is required to conduct tests to confirm its compatibility with the launch vehicle environment. The results of these tests must be reported to the launch vehicle provider. The required tests, for the Qualification Model, Protoflight Model, and Flight Model are shown in Table 6. (Note: DLF refers to Design Load Factor and CLA refers to Coupled Load Analyses)

Table 6. Satellite Structural Tests, Margin, and Duration

Test	Qualification Model	Protoflight Model	Flight Model
Static Load			
Level	1.25 x Limit	1.25 x Limit	1.0 x Limit
Analyses	DLF or CLA	DLF or CLA	DLF or CLA
Acoustic Environment			
Level	Limit + 3dB	Limit + 3dB	Limit Level
Test Duration	2 Minutes	1 minute	1 minute
Sine Wave Vibration			
Level	1.25 x Limit	1.25 x Limit	1.0 x Limit
Sweep Rate	2 Octaves/Minute	4 Octaves/Minute	4 Octaves/Minute
Shock Impact			
Number of Times	2 Times	2 Times	1 Time

Integration

The satellite developer can conduct a fit check of the separation system 8 months prior to launch.

Access to the satellite ends at 15 hours prior to liftoff, when the doors of the fairing are closed. The battery must be charged for 50 hours of power, in order to prepare for a possible one-day postponement of liftoff due to the primary payload.

Lead Time Required

A set of information about the piggyback satellite must be forwarded to the launch vehicle provider 15 months prior to launch, to enable the launch vehicle provider to prepare for such launch activities as Flight Analysis, Change of Rocket Specification, etc.

The specific information that needs to be provided includes the following:

- Purpose of the satellite, mission, and desired orbit
- Satellite development plan (schedule, test plan, etc.)
- Satellite mass properties
- Special requirements for environmental conditions, interface conditions, and work at launch site

Milestone Schedule

The following reviews are planned prior to launch:

- ***Mission Analysis Review***
  Time: 3 months prior to launch
  Purpose: To confirm if the launch vehicle meets the interface requirements for flight

- ***Satellite Interface Review***
  Time: Just before delivery of satellite to launch site
  Purpose: To confirm if the satellite meets the interface requirements for flight

- ***Launch Preparation Review***
  Time: 4 to 7 days prior to launch
  Purpose: To confirm launch preparation status

Pricing

The price to the small, piggyback satellite developers is based on the cost of the safety analysis, any special requirements (e.g., command signals, photographing, telemetry), and the share of the launch vehicle weight.

Concluding Remarks

A website for small satellite developers is being prepared. Just as there are a variety of small satellites, their requirements of the launch vehicle vary greatly in complexity as well. Therefore, those requirements are to be carefully reviewed by both the small satellite developer and MHI for its successful launch.

Acknowledgments

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References

1. NASDA Mission Operations Department, H-IIA User's Manual, NASDA.